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SUMMARY

A pilot-model analysis has been performed that relates pilot control compensation, pilot-aircraft system response characteristics, and aircraft response characteristics for longitudinal control. The results show that a higher aircraft short period frequency is required to achieve superior pilot-aircraft system response in an altitude control task than is required in an attitude control task. These results were then compared with those of a simulation study in which the dynamics of a simulated F-8C research aircraft were studied with two different control systems. One of these systems used blended pitching velocity and normal acceleration feedback signals in the elevator control loop with gains that provided a relatively low short period frequency. The other control system was a damping augmentation system with a gain on pitch rate that provided a high short period frequency. In this study the pilots preferred the system with the high short period frequency for short range tracking of a target aircraft. The pilot-model analysis and the simulation study were, therefore, in agreement.

It was concluded that the pilot-model analysis provides a theoretical basis for determining the difference in difficulty for the pilot between altitude control tasks (for example, short range tracking, formation flying, and glide slope following) and attitude control tasks (such as cruise flight and long range tracking).

INTRODUCTION

The Cooper-Harper pilot rating scale provides a measure of the pilot difficulty in performing any given task, but it provides no recognition of the relative difficulty of different tasks. The pilot-model analysis of reference 1 provides a means of distinguishing the difficulty for the pilot of performing attitude control tasks and altitude control tasks. These differences in difficulty cover the differences between specific tasks involving attitude control, such as cruise flight and long range tracking, and tasks involving altitude control, such as short range tracking, formation flying, and path following tasks. The present investigation extends the analysis of reference 1 to further differentiate between these two classes of control tasks. The present analysis allows a better understanding of the results of handling quality tests that involve different control tasks.

The results of the pilot-model analysis have been compared with the results of a simulation study of two different digital control systems implemented on a simulated F-8C research aircraft. In this simulation study, pilot opinions were obtained for each of the two control systems for control tasks that involved both short range tracking of a target aircraft and attitude regulation without a target.

The results were also compared with the results of reference 2, in which air combat maneuvering was performed with a variable-stability aircraft. Various aircraft configurations were used in tests both with and without a target aircraft in reference 2 so that data on pilot ratings for different aircraft responses for different control tasks were provided.

SYMBOLS

Values are given in SI Units. Measurements were made in U.S. Customary Units.

b	wing span, m
C*	$= N_z + 0.175q$, g units
C_D	$= \dfrac{\text{Drag}}{\bar{q}S}$
$C_{D_{SB}}$	$= \dfrac{\partial C_D}{\partial \text{ Speed brake}}$
C_L	$= \dfrac{\text{Lift force}}{\bar{q}S}$
C_{L_α}	$= \dfrac{\partial \text{ Lift force}}{\partial \alpha} \dfrac{1}{\bar{q}S}$
$C_{L_{SB}}$	$= \dfrac{\partial C_L}{\partial \text{ Speed brake}}$
C_{l_p}	$= \dfrac{\partial \text{ Rolling moment}}{\partial p} \dfrac{2V}{b^2\bar{q}S}$
C_{l_r}	$= \dfrac{\partial \text{ Rolling moment}}{\partial r} \dfrac{2V}{b^2\bar{q}S}$
C_{l_β}	$= \dfrac{\partial \text{ Rolling moment}}{\partial \beta} \dfrac{1}{\bar{q}Sb}$
$C_{l_{\delta_a}}$	$= \dfrac{\partial \text{ Rolling moment}}{\partial \delta_a} \dfrac{1}{\bar{q}Sb}$

$$C_{l_{\delta_r}} = \frac{\partial \text{ Rolling moment}}{\partial \delta_r} \frac{1}{\bar{q}S_b}$$

$$C_m = \frac{\text{Pitching moment}}{\bar{q}S\bar{c}}$$

$$C_{m_q} = \frac{\partial \text{ Pitching moment}}{\partial q} \frac{2V}{\bar{c}^2\bar{q}S}$$

$$C_{m_\alpha} = \frac{\partial \text{ Pitching moment}}{\partial \alpha} \frac{1}{\bar{c}\bar{q}S}$$

$$C_{m_{\delta_e}} = \frac{\partial \text{ Pitching moment}}{\partial \delta_e} \frac{2V}{\bar{c}^2\bar{q}S}$$

$$C_{m_{\dot{\alpha}}} = \frac{\partial \text{ Pitching moment}}{\partial \dot{\alpha}} \frac{2V}{\bar{c}^2\bar{q}S}$$

$$C_{n_p} = \frac{\partial \text{ Yawing moment}}{\partial p} \frac{2V}{b^2\bar{q}S}$$

$$C_{n_r} = \frac{\partial \text{ Yawing moment}}{\partial r} \frac{2V}{b^2\bar{q}S}$$

$$C_{n_\beta} = \frac{\partial \text{ Yawing moment}}{\partial \beta} \frac{1}{\bar{q}Sb}$$

$$C_{n_{\delta_a}} = \frac{\partial \text{ Yawing moment}}{\partial \delta_a} \frac{1}{\bar{q}Sb}$$

$$C_{n_{\delta_r}} = \frac{\partial \text{ Yawing moment}}{\partial \delta_r} \frac{1}{\bar{q}Sb}$$

$$C_{Y_p} = \frac{\partial \text{ Side force}}{\partial p} \frac{2V}{\bar{q}Sb}$$

$$C_{Y_r} = \frac{\partial \text{ Side force}}{\partial r} \frac{2V}{b\bar{q}S}$$

C_{Y_β} $= \dfrac{\partial \text{ Side force}}{\partial \beta} \dfrac{1}{\bar{q}S}$

$C_{Y_{\delta_a}}$ $= \dfrac{\partial \text{ Side force}}{\partial \delta_a} \dfrac{1}{\bar{q}S}$

$C_{Y_{\delta_r}}$ $= \dfrac{\partial \text{ Side force}}{\partial \delta_r} \dfrac{1}{\bar{q}S}$

$\bar{c}$ mean aerodynamic chord, m

$\left.\begin{matrix} F_1, F_4, F_5, \\ F_6, F_7, F_8 \end{matrix}\right\}$ forward loop static gains

g acceleration of gravity, m/sec^2

h altitude, m

I_Y moment of inertia, $kg\text{-}m^2$

K control system gain

K_h pilot-model static gain in altitude control, rad/m

K_q, K_p, K_r stability augmentation gains, per sec

K_θ, K_ϕ stability augmentation gains, rad/rad

K_θ' pilot-model static gains in pitch control

L_α normalized lift-force derivative, $\dfrac{\rho VS}{2m} C_{L_\alpha}$, per sec

M Mach number

M_q normalized damping in pitch, $\dfrac{\rho VS\bar{c}^2}{4I_Y} C_{m_q}$, per sec

M_α normalized pitching-moment derivative, $\dfrac{\rho V^2 S\bar{c}}{2I_Y} C_{m_\alpha}$, per sec^2

M_{δ_e} normalized pitching moment, $\dfrac{\rho V^2 S\bar{c}}{2I_Y} C_{m_{\delta_e}}$, per sec^2

m	mass, kg
N_y, N_z	normal and lateral accelerations, g units
p, q, r	rolling, pitching, and yawing angular velocity, rad/sec or deg/sec
$\bar{q}$	dynamic pressure, $\rho V^2/2$, N/m^2
S	wing area, m^2
s	Laplace operator, per sec
T	time constant, sec
T_1, T_2	pilot-model lag and lead time constants, respectively, sec
u	nondimensional airspeed
V	velocity, m/sec
x, y	distance along body axis, m
α	angle of attack, rad or deg
β	angle of sideslip, rad or deg
γ	flight-path angle, rad or deg
δ	control surface deflection, rad or deg
$\delta_a, \delta_r, \delta_e$	aileron, rudder, and elevator control deflections, rad or deg
ζ	damping ratio
θ, ψ, ϕ	pitch, yaw, and roll angles, rad or deg
$\lambda, \lambda_1, \lambda_2$	real roots, rad/sec
ρ	air density, kg/m^3
ω	frequency, rad/sec

Subscripts:

c	command
e	error
o	output

sp short period

Dot over symbol denotes derivative with respect to time.

DESCRIPTION OF ANALYTICAL AND SIMULATION STUDIES

Pilot-Model Analysis

The pilot model used in the analysis is based on measurements reported in reference 3. In this reference pilot response was measured by using a model matching, parameter tracking method. The model form used was

$$\frac{\text{Output}}{\text{Input}} = \frac{K_\theta'(1 + T_2 s)}{(1 + T_1 s)^2}$$

The parameters K_θ', T_1, and T_2 were automatically adjusted to provide the best least-squares match to the subject's input-output time history during a single axis pitch compensatory tracking task. In those tests the subject controlled various simplified vehicle dynamics which contained lags which spanned the lag to be found in aircraft response. The tests showed that as the lag contained in the vehicle is increased from that contained in a pure rate command system K/s, the pilot compensated by increasing his lead time constant T_2 from 0 to 1 sec. Further increase in vehicle lag resulted in further compensation in the form of a decrease in lag time constant T_1 from 0.2 to 0.04 sec. The tests and measurements in reference 1 determined that the preferred pilot response included zero lead ($T_2 = 0$ sec) and a lag time constant (T_1) of 0.2 sec. They also determined that the pilot adjusted his static gain K_θ' to obtain a pilot-aircraft system frequency around 2.5 rad/sec. This pilot model, being linear, provides a description of pilot response which is very convenient to use in analytical studies.

In reference 1, a study was made which related those pilot-model parameters to experience that has been accumulated over many years with respect to the aircraft longitudinal handling qualities. The study showed good correlations with pilot ratings when the pilot model was used in conjunction with the following criteria for pilot model-aircraft system pitch response characteristics: oscillatory frequency of 2.5 rad/sec with zero damping and a real root of -0.4 rad/sec. That is, using the preferred pilot response with $T_2 = 0$ and $T_1 = 0.2$ sec, a pilot model-aircraft system response of $\omega_\alpha = 2.5$ rad/sec, $\zeta_\alpha = 0$, and $\lambda_\theta = -0.4$ rad/sec could be obtained with those aircraft which were rated satisfactory and could not be obtained with those aircraft which were rated acceptable.

In addition to the study of pitch attitude control, reference 1 contains an investigation of altitude control. A multiloop pilot model shown in figure 1 was used in the study of altitude control. For this multiloop pilot model, a specification of system response of two oscillatory modes of motion, $\omega_\alpha = 2.5$ rad/sec and $\zeta_\alpha = 0$, and $\omega_h = 1.2$ rad/sec and $\zeta_h = 0$, provided good correlation with pilot ratings obtained in a glide slope following task.

The study showed that there were differences in the aircraft response characteristics required to meet the altitude control specifications and the attitude control specifications. It was concluded that there were fundamental differences between altitude and attitude control tasks.

In making these calculations in reference 1, the aircraft was represented by the two-degree-of-freedom linear equations:

$$\dot{\alpha} = q - L_{\alpha}\alpha$$

$$\dot{q} = M_q q + M_{\alpha}\alpha + M_{\delta_e}\delta_e$$

and the relationship for altitude

$$\dot{h} = V(\theta - \alpha)$$

The aerodynamic stability derivatives used in these equations are related to the aircraft short period frequency and damping by the equations:

$$\omega_{sp}^2 = -L_{\alpha}M_q - M_{\alpha}$$

$$2\zeta_{sp}\omega_{sp} = L_{\alpha} - M_q$$

In the present investigation, additional pilot model-aircraft system calculations are made to examine the effect of specifying improved system damping. Those aircraft for which a system response of $\omega_{\alpha} = 2.5$ rad/sec, $\zeta_{\alpha} = 0.1$, and $\lambda_{\theta} = -0.4$ rad/sec for pitch control and $\omega_{\alpha} = 2.5$ rad/sec, $\zeta_{\alpha} = 0.1$, $\omega_h = 1.2$ rad/sec, and $\zeta_h = 0.1$ for altitude control could be obtained were determined by using the same pilot model of preferred pilot response ($T_1 = 0.2$ sec, $T_2 = 0$ sec). A system response with damping ratios of 0.1 would undoubtedly be preferred by pilots over a system response with no damping. Of course, it is to be expected that achieving this improved system response would require improved aircraft response, that is, a higher aircraft short period frequency and/or a higher damping ratio. It is believed that these additional calculations provide insight regarding the sensitivity of aircraft response requirements to the change in system response specifications that is useful in the interpretation of the simulator results.

These calculations were performed by selecting a typical high speed value for L_{α} ($L_{\alpha} = 1.3$) and a number of values for aircraft short period frequency and damping ratio. The corresponding values for the stability derivatives m_q and M_{α} were then determined. These stability derivatives were then used in the aircraft equations of motion together with the pilot model, and the pilot model-aircraft system response characteristics were determined. To illustrate the results, selected test points and the system characteristics for altitude control are shown in table I.

The comparison between the computed results and the simulator results rests on the assumption that if the aircraft's short period characteristics are the same and the values of L_{α} are nearly equal, then the aircraft appears the same to the pilot. In the calculations the short period characteristics are derived from the required values of aerodynamic stability derivatives, as explained earlier. In the simulation study the short period characteristics were obtained through stability augmentation.

Simulation Study

The simulation study was conducted on the Langley differential maneuvering simulator. Data for the handling qualities of two proposed control systems were obtained. Resulting data were compared to the results of the pilot-model analysis. Descriptions of the aircraft mathematical model, the two control laws studied, the aircraft-control system response characteristics, and the pilot tasks follow.

Aircraft mathematical model.- The F-8C aircraft was represented by six-degree-of-freedom, nonlinear equations of motion similar to those presented in reference 4. Lift, drag, and pitching-moment aerodynamic data were included as functions of Mach number, angle of attack, and control deflection. Other aerodynamic coefficients were included with less elaborate functions. Typical aerodynamic data for a Mach number of 0.6 are shown in table II. These data were obtained from reference 5. The coefficients used included effects of aircraft flexibility. Control actuators were represented by first order lags with rate and displacement limits. This aircraft was combined with two different control laws which are designated control law 1 and control law 2.

Control law 1.- A block diagram of the longitudinal augmentation for control law 1 is shown in figure 2. This control law is a C* arrangement which was first proposed in reference 6. In addition, control law 1 is discussed specifically in reference 7. With this arrangement the pilot commands a blend of normal acceleration and pitching velocity, $N_Z + 0.175q$ (where N_Z is in g units and q is in deg/sec). Forward loop integration is also provided by a low frequency cancelling signal of the actuator position feedback signal. A derivation of this function is given in appendix A. The forward loop static gain F_1 is scheduled as a function of dynamic pressure. A conventional center control stick was used. Elevator stick force was provided by a nonlinear feel spring whose force characteristics were similar to the feel system used on the production F-8C. (See fig. 3.) A small dead band and a constant gearing gain of 0.45 g/cm were applied to the stick position signal and resulted in the force sensitivity characteristics shown in figure 4. The ordinate scale in this figure is the C* command value.

For lateral control the control law contained rolling velocity in the aileron control loop; rolling velocity, yawing velocity, lateral acceleration, and an aileron-to-rudder crossfeed were contained in the rudder control loop. (See fig. 5.) The variation of aileron stick displacement with force was nonlinear with a soft stop arrangement. (See fig. 6.) Forward loop gains on the lateral stick input to the system were a function of angle of attack, and each gain's variation with angle of attack is shown in figure 5. Rudder control included a constant force gradient and was linear. A more detailed description of the complete system is presented in reference 7.

Control law 2.- The second control law was a damping augmentation system. In the longitudinal control loop, pitching velocity was the nominal feedback signal. When the elevator control stick was in its center dead band, an additional feedback signal, the integral of pitching velocity, was used. Integration of pitching velocity began when the elevator control stick entered the dead band and was reset to zero when the elevator control stick moved outside the dead band. The system therefore provided a rate command, attitude hold function. A block diagram is shown in figure 7. Static forward loop gains were scheduled as a function of Mach number and altitude and are shown in table III. A linear interpolation scheme was used in conjunction with this table to provide a continuous function. The control stick used with this control system was a semirigid, side-arm, force input stick. The control stick's longitudinal output was commanded pitching velocity with the nonlinear variation shown in figure 8(a). This type of control stick was selected for use with control law 2 because of the favorable report given for a semirigid, force input, side-arm controller in reference 8 and for a side-arm controller given in reference 9. A more conventional design approach was taken for control law 1, and the standard center stick was selected for use with control law 1.

The lateral control system contained the aileron and rudder control loops with the aileron loop being similar to the longitudinal loop. Rolling velocity feedback was nominally used in the aileron loop with the integral of rolling velocity added when the aileron control stick was in the control dead band. A yawing velocity feedback with a washout filter was used in the rudder control loop. Forward loop gains for the aileron and rudder control loops were functions of Mach number and altitude as shown in table III. Stick output was commanded rolling velocity with the nonlinear variation shown in figure 8(b). Rudder pedal output was linear with no dead band.

Aircraft-control system response.- The steady state longitudinal and lateral responses of the F-8C aircraft with control law 1 and control law 2 are given in table IV. These results were obtained with the full, nonlinear equations of motion. Steady state pitching velocity and peak values of normal acceleration that result from a step input of 1/2 full stick deflection (corresponding to approximately 48 N stick force) for control law 1 and a step input of 22 N for control law 2 for two flight conditions are shown. Also shown is steady state rolling velocity for lateral step inputs of 1/2 full deflection of the center stick and 22 N for the side stick, where these lateral responses were obtained for flight at 1g. Table IV shows that for control law 1 the stick sensitivity was such that the force per g was 27 and 32 N for the two flight conditions. For control law 2 the corresponding values were 7.5 and 8.4 N per g. Since the stick output was a nonlinear function of force, those values of N per g apply only for the value of input given. From these values control law 2 appears to have the greater steady state sensitivity. However, since control law 1 used a center stick and control law 2 used a side stick, a direct comparison is not possible.

Table V presents the dynamic response characteristics of the two systems at the same two flight conditions used in table IV. These dynamic response characteristics were derived from linear system equations such as those presented in appendix B. An examination of table IV shows that the longitudinal dynamic response characteristics of the aircraft plus control law 1 have lower short period frequencies than do those of the aircraft plus control law 2. This difference is further illustrated in figure 9, where simulator time histories of the system responses to step stick inputs of 1/5 full stick deflection for system 1 and 0.45 N for system 2 are shown. It can be seen that control law 1 brings about a response with a large pitching velocity overshoot, which exists for a relatively long time because of the low short period frequency.

The difference in the structure of the two control laws also causes a difference in the roots associated with the phugoid mode. Initially, it was hoped that the large negative real root, $\lambda = -0.436$ rad/sec, brought about by control law 2 when the stick was inside the dead band would result in very rapid settling on the final steady state value when changing pitch attitude, which would benefit precise maneuvering. However, simulation tests of control law 2 both with and without feedback of the integral of pitching velocity signal when the stick was inside the dead band showed that this large real root was not a significant factor in the tracking task used in this study. It was concluded that the most significant difference in the two systems was the difference in short period frequency. The pilot-model analysis results are also concerned with short period frequency, and results of the two separate analyses were compared.

The lateral responses of the two systems were more nearly the same than were the longitudinal responses. A better damped, higher frequency Dutch roll mode was obtained with control law 1 than with control law 2. A faster roll mode of motion was obtained with the second control law (a real root of $\lambda = -9.6$ rad/sec for control law 2 as compared with a nearly critically damped oscillation $\omega = 5.65$ rad/sec, $\zeta = 0.96$ for control law 1). The steady state roll responses by the two systems with approximately comparable control inputs were nearly the same. These differences were small and should not cause differences in the handling qualities evaluations of the two systems. Only the longitudinal characteristics are examined in detail in this report.

Pilot tasks.- As previously mentioned, this study was conducted using the Langley differential maneuvering simulator whose operation is described in reference 10. This simulator can be used to study one pilot aircraft system in competition with another in an air combat situation. This is accomplished by computing the relative position and attitude of the two aircraft and displaying this information to each of the two pilots, who are located in two separate spheres. The spheres provide an all-around projection screen. In the present study only one sphere was used together with a programmed target time history. In this way the same task was presented with each of the two control laws being investigated, and a quick comparison was obtained.

Two research pilots took part in this study. Each was a military pilot temporarily attached to NASA, and each had fighter experience. These men performed a series of four increasingly difficult tasks in evaluating the handling qualities of the two systems. They first performed an open-loop task in which they closed the loop only on stick deflection by moving the stick to a reference position of their choice and by observing the response of the aircraft. Next, they closed an attitude loop, controlling the aircraft to an attitude reference of their choice. Next, a target aircraft flying straight and level was used as a position reference. Initial conditions were established at a range of 200 m with the aircraft displaced from the flight path of the target a few meters either vertically or laterally. The pilot then maneuvered the aircraft to a position in-trail behind the target as rapidly as possible. Finally, they attempted to follow a maneuvering target which was provided with the time history shown in figure 10. This time history was made with the F-8C and control law 2 and was stored on tape for repeated use. It can be seen that the maneuvers were not extremely violent. No excessive values of normal acceleration were used, and periods of straight and level flight occurred throughout the run. The run started at a condition of $M = 0.80$ at an altitude of 6000 m, and airspeed increased to $M = 0.95$ during the run. The subject aircraft was initially positioned at a range of 200 m directly behind the target. The range usually increased to 600 m at various times during the run.

RESULTS

Pilot-Model Analysis

The analysis which used the pilot model was performed to obtain information on the difficulty of achieving a specified set of pilot-model aircraft system response characteristics by determining the aircraft response characteristics required to combine with the pilot model to achieve the specified system response. The basic hypothesis in the calculations is that the pilot would prefer to act as a very simple controller, which in the model is expressed by the use of a lag time constant of 0.2 sec and a lead time constant of 0 sec, while at the same time achieving a superior system (pilot plus aircraft) response. The pilot-model static gain is considered a free variable in these computations because it has been shown that so long as the control stick deflections are not extreme (neither extremely small nor extremely large) the pilot can freely adjust his static gain. Data presented in reference 3 show that a reduction in stick sensitivity to one-half of the nominal value results in no change in original system frequency, and a reduction in sensitivity to one-tenth nominal results in a reduction in system frequency of only one-half of the original value.

By using the pilot model described above, it is possible to determine unique combinations of aircraft short period frequency and damping factor that provide specified system characteristics. One set of specifications for longitudinal pitch attitude response that has been shown to be useful is a real root greater in magnitude than -0.4 rad/sec and a short period frequency greater than 2.5 rad/sec with a damping ratio of 0. For altitude control, a specification of altitude mode with motion frequency greater than 1.2 rad/sec is added. These specifications have been shown to define boundaries of aircraft response that agree with the handling qualities boundaries for pilot ratings of 3.5 (satisfactory). These results are presented in reference 2. Reference 2 also shows that the boundaries for altitude control and attitude control are different. These curves are shown in the lower left-hand area of figure 11.

To obtain information on the sensitivity of the required aircraft response characteristics to changes in the specified system characteristics, the specified damping factors were changed from 0 to 0.1, and a new set of boundaries were computed (see fig. 11). These new boundaries were drawn so that their shape corresponded to the shape determined for the boundaries in reference 2, with their locations determined by an approximate interpolation of the data points presented in table I. Although this method of locating the boundaries is not as accurate as the method used in reference 2, it is accurate enough to indicate that increasing the damping requirement from 0 to 0.1 does move the boundaries to a region of higher aircraft short period frequency and damping. It can also be seen that the difference between the boundary for altitude control and the boundary for attitude control is even greater than in the previous results. It can be concluded from these curves that a higher aircraft short period frequency and damping are required to obtain a desirable altitude control than those which are required to obtain an equally desirable attitude control. They further indicate accentuation of this difference for the 0.1 damping requirement. The main purpose of the present calculation was to illustrate this difference.

The relationship between these computed pilot model-aircraft system response characteristics and the tasks performed in the simulation study is as follows. In the simulation tasks with no target, the only requirement placed on the pilot was to control attitude; therefore, those tasks can be compared with the analysis of attitude control. In the simulation tasks with a target, the pilot had to control altitude if the range was short. The range at which altitude control is required is illustrated with the following computation. The pilot model altitude loop gain K_hV varied from 0.3 to 3.0 rad/sec in table I. A value of 2.0 would be approximately correct for the conditions of the simulation tests to have a stable altitude response with the aircraft used in the tests. The value for V was approximately 250 m/sec so that the value of commanded pitch angle per meter of displacement K_h would be 0.008. For a tracking aircraft vertically displaced 30 m from the flight path of the target and traveling in the same direction as the target, and for a pilot closing the altitude loop with this gain, the range at which the aircraft would point at the target may be found as follows:

$$\Delta\theta = K_h \, \Delta h = (0.008)(30) = 0.24 \text{ rad}$$

$$\text{Range} = \Delta h \cot (0.24) = 125 \text{ m}$$

If the target were at a range of less than 125 m and the pilot attempted to point directly at the target, the gain K_h would effectively be greater than 0.008 rad/m, and an unstable or poorly damped position response would result. This example illustrates that at a range of around 125 m, the pitch angle that the pilot can command is determined by the requirement for a stable position response rather than by the desire to point directly at the target. The example further illustrates that the aircraft can be pointed steadily at the target only by positioning the aircraft directly behind the target. As the range increases, the value of the pitch angle required to point at the target becomes less than that dictated by a stable position response; therefore, the task becomes primarily an attitude control task.

The gain K_h used in computing the range noted in the preceding equations was based on the pilot-model analysis which determined maximum gains. The gain that the pilot might actually use could be smaller; therefore, the range at which the tracking task was an altitude control task would be larger than 125 m.

Also, the gain is a function of aircraft response characteristics and velocity; therefore, it varies with flight conditions. The simulation task of tracking the maneuvering target which was started with a range of 200 m was seen predominately as an altitude (or position) control task.

EVALUATION OF SIMULATED FLIGHT CONTROL SYSTEMS

The longitudinal characteristics of the simulated F-8C aircraft with the two flight control systems studied, is shown at two flight conditions in table IV. System 1, a C*-type system, had lower short period frequency and slightly higher damping ratio. System 2, a pitch rate damping system, had higher short period frequency and slightly lower damping.

The two pilots that participated in the study both preferred system 1 for the attitude control task when the horizon was used as the reference, that is, no-target aircraft. The pilots believed that the damping augmentation system provided by control law 2 was too sensitive and that the response was too abrupt.

The pilot-model analysis does not explain why control law 1 is preferred to control law 2 for attitude control tasks. Both systems are well within the boundary for superior attitude control shown in figure 11.

When the target aircraft was used, either flying straight and level or maneuvering, the pilots preferred control law 2. With the target there was no complaint that the response was too abrupt. Control law 1 was judged to be too slow in response and the control force required to be too high. These pilot comments apply principally to the longitudinal response. In lateral control the two systems were rated equal with perhaps a slight advantage going to control law 1.

The pilots' preference for control law 2 in the altitude control (tracking) task is consistent with results of the pilot-model analysis. The pilot-model analysis, used to define the boundaries in figure 11, indicated that superior altitude control would be expected with a high aircraft short period frequency. The pilot-model analysis, therefore, predicts that control law 2 should be preferred to control law 1 for the tracking task. This result is not meant to suggest that the C* arrangement of control law 1 is at fault; it indicates only that the gains used in control law 1 are such that the law would not be preferred to control law 2.

Figures 12 and 13 show time histories of tracking the maneuvering target, which started from a position directly behind the target, for each control law. Based on the performance displayed in these figures, the two systems must be judged equal. For example, the altitude control that occurs at the 40-sec mark in the run appears to be controlled with better damping with control law 2 than with control law 1; however, the overall amplitudes of the position errors Δh and Δy appear to be about equal for the two systems. A task with fewer variables, in which the differences between the two systems can be seen more clearly, is that of moving to a position behind a target flying straight and level from a displaced position. Time histories of this type of maneuver with the two systems are shown in figure 14. From this figure it can be seen that with control law 2 the change in position is accomplished more rapidly; that is, the altitude mode of motion has a shorter period.

The results of reference 2, which involved flight tests with a variable-stability aircraft, are also in agreement with the present results. In reference 2, tests were made both with and without a target aircraft. In these tests the range to the target aircraft varied from around 600 m to 150 m. With a high short period frequency (ω_{sp} = 10 rad/sec; ζ_{sp} = 0.44), the pilots rated the configuration satisfactory (pilot rating of 3) when a target was used, and unacceptable (pilot rating of 8) with no target. With a low frequency configuration (ω_{sp} = 3; ζ_{sp} = 0.68) the pilot ratings were 6 with no target and 7.5 with the target. These results agree with the results of the present study in that the high frequency airplane is preferred for target tracking.

CONCLUDING REMARKS

A simulation study of two different control systems combined with the dynamics of an F-8C aircraft has shown that the control system that provided the higher short period frequency was preferred for short range target tracking tasks (altitude control tasks) over the control system which provided a low short period frequency. Conversely, pilots preferred the lower short period frequency for a pitch attitude control task using the horizon for reference.

Pilot preference for the high short period frequency system for the tracking task was anticipated based on the use of a pilot model to predict the region of improved system response. Although the pilot-model analysis did not predict the pilots' preference for the attitude control task, it did demonstrate that pilot-model analysis can be a useful adjunct in flight control system evaluation.

The results emphasize the importance of recognizing the class of control task, whether altitude control or attitude control, in analyzing or rating flight control systems. The class of control task must be recognized to ensure the proper interpretation of pilot ratings.

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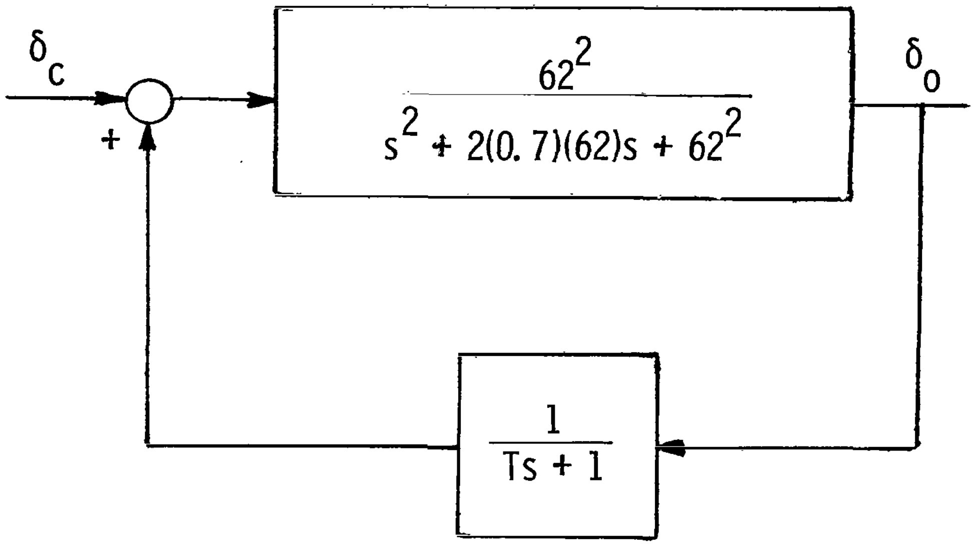
APPENDIX A

SERVO INTEGRATION DERIVATION

A derivation of the servo operation for the elevator control of control system 1 is as follows. The servo subsystem can be represented as shown in the sketch.



The basic servo and the mechanical feedback is represented by the forward block, and the destabilizing cancelling signal is the feedback block. The input to output transfer function for the subsystem is:

$$\frac{\delta_o}{\delta_c} = \frac{\dfrac{62^2}{s^2 + 2(0.7)(62)s + 62^2}}{1 - \left[\dfrac{62^2}{s^2 + 2(0.7)(62)s + 62^2}\right]\left(\dfrac{1}{Ts + 1}\right)}$$

$$= \frac{62^2(Ts + 1)}{Ts\left\{s^2 + s\left[\dfrac{1}{T} + 2(0.7)(62)\right] + 62^2 + \dfrac{2(0.7)(62)}{T}\right\}}$$

APPENDIX A

The low frequency response is accurately represented by neglecting the high frequency term $\left\{s^2 + s\left[\frac{1}{T} + 2(0.7)(62)\right] + 62^2 + \frac{2(0.7)(62)}{T}\right\}$, so that

$$\frac{\delta_o}{\delta_c} = \frac{Ts + 1}{Ts}$$

This expression shows that a proportional plus integral function is provided by the system

$$\frac{\delta_o}{\delta_c} = 1 + \frac{1}{Ts}$$

APPENDIX B

CLOSED-LOOP SYSTEM CHARACTERISTICS

The closed-loop system response characteristics were determined from linear equations such as those given below. This particular example is for the longitudinal response of the aircraft plus control system 2 for a flight condition of $M = 0.67$ at an altitude of 6100 m.

$$\dot{q} = -0.4877q - 4.790\alpha - 8.743\delta_e$$

$$\dot{u} = -0.0148u - 13.87\alpha - 32.2\theta$$

$$\dot{\alpha} = q - 0.836\alpha - 0.1115\delta_e$$

$$\dot{\theta} = q$$

$$\dot{\delta}_e = 12.5K_q + 12.5K_\theta - 12.5\delta_e$$

where $K_q = 0.7$ and $K_\theta = 1.43$ for these flight conditions.

Further information on the matrices used to represent the airplane plus control system 1 can be found in reference 7.

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TABLE I.- ALTITUDE CONTROL AIRCRAFT - PILOT-MODEL SYSTEM CHARACTERISTICS

Aircraft parameters			Pilot-model gains		Closed-loop characteristics					
L_α, per sec	ω_{sp}^2, (rad/sec)^2	$2\zeta_{sp}\omega_{sp}$, rad/sec	K_θ'	K_hV, rad/sec	ω_h, rad/sec	ζ_h	ω_α, rad/sec	ζ_α	ω_δ, rad/sec	ζ_δ
1.3	5	3	6	0.3	0.55	0.82	2.22	0.11	6.15	0.94
1.3	5	5	10	.4	.77	.77	2.13	.15	7.07	.93
1.3	20	3	16	2.0	1.17	.10	4.10	.10	6.55	.91
1.3	20	5	30	2.5	1.60	.10	3.87	.10	7.85	.89
1.3	100	3	60	2.0	1.04	.11	9.35	.11	6.40	.84
1.3	100	5	100	2.0	1.25	.15	8.60	.17	7.63	.76
1.3	100	10	200	3.0	1.75	.14	6.90	.13	11.50	.77

TABLE II.- TYPICAL F-8C AERODYNAMIC DATA; M = 0.6

δ_e	Pitching-moment coefficient, C_m, for α of -												
	-8	-4	0	4	8	10	12	14	16	18	20	22	24
-25	0.300	0.290	0.280	0.265	0.245	0.234	0.220	0.200	0.181	0.170	0.166	0.150	0.080
-20	.283	.262	.242	.224	.198	.180	.160	.140	.120	.103	.101	.060	0
-15	.256	.222	.196	.169	.142	.124	.100	.074	.046	.026	.024	-.010	-.050
-5	.130	.096	.064	.032	-.005	-.030	-.054	-.082	-.110	-.131	-.133	-.160	-.180
0	.058	.024	.010	-.044	-.083	-.110	-.134	-.166	-.190	-.196	-.199	-.220	-.250
5	-.025	-.060	-.095	-.127	-.164	-.196	-.210	-.231	-.255	-.275	-.276	-.290	-.330

δ_e	Lift coefficient, C_L, for α of -												
	-8	-4	0	4	8	10	12	14	16	18	20	22	24
-25	-0.770	-0.510	-0.240	0.003	0.283	0.408	0.510	0.595	0.688	0.780	0.840	0.784	0.697
-20	-.740	-.485	-.220	.040	.320	.442	.555	.660	.750	.840	.890	.834	.747
-15	-.720	-.460	-.180	.070	.355	.484	.600	.711	.804	.880	.950	.894	.807
-5	-.630	-.370	-.090	.182	.466	.610	.712	.820	.920	.990	1.020	.964	.877
0	-.580	-.320	-.032	.234	.520	.660	.770	.880	.978	1.040	1.070	1.014	.927
5	-.529	-.256	-.030	.297	.580	.720	.828	.940	1.030	1.100	1.130	1.074	.987

TABLE II.- Continued

Coefficients	Angle of attack of -												
	-8	-4	0	4	8	10	12	14	16	18	20	22	24
C_D	0.083	0.030	0.013	0.030	0.058	0.090	0.141	0.201	0.260	0.322	0.382	0.942	0.501
C_{l_p}	-.332	-.330	-.311	-.333	-.288	-.226	-.182	-.189	-.181	-.124	-.108	-.159	-.162
C_{l_r}	-.101	-.004	.005	.015	.029	.045	.066	.081	.097	.125	.133	.124	.135
C_{l_β}	-.034	-.053	-.080	-.129	-.154	-.141	-.124	-.111	-.131	-.207	-.156	-.143	-.144
$C_{l_{\delta_a}}$	.049	.049	.048	.051	.051	.046	.038	.029	.024	.022	.022	.022	.022
$C_{l_{\delta_r}}$	.010	.017	.025	.025	.026	.026	.026	.024	.027	.031	.034	.037	.041
C_{n_p}	.014	.033	.031	-.019	-.029	-.016	-.005	.004	.011	.023	.010	-.032	-.022
C_{n_r}	-.306	-.308	-.306	-.306	-.306	-.303	-.327	-.383	-.424	-.443	-.443	-.461	-.457
C_{n_β}	.222	.168	.130	.176	.188	.174	.161	.144	.111	.059	.047	.047	.042
$C_{n_{\delta_a}}$	.005	.009	.012	.014	.014	.013	.011	.009	.009	.009	.009	.010	.010
C_{Y_p}	-.102	-.155	-.059	.042	.122	.086	.093	.126	.132	.153	.046	.161	.177
C_{Y_β}	-1.02	-1.02	-1.02	-1.04	-1.09	-1.16	-1.16	-1.20	-1.23	-1.21	-1.21	-1.21	-1.21
$C_{Y_{\delta_a}}$	.023	.023	.023	.017	.011	.008	.006	.003	0	.003	-.006	-.006	-.006

TABLE II.- Concluded

Constant aerodynamic data with respect to δ_e and α

C_{m_q}	-4.0
$C_{m_{\dot{\alpha}}}$	-0.35
$C_{n_{\delta_r}}$	-0.103
$C_{Y_{\delta_r}}$	0.199
C_{Y_r}	0.398
$C_{D_{SB}}$	0.11
$C_{L_{SB}}$	0.0003

Altitude, m	Military rated thrust for velocity, m/sec, of -				
	30.5	122	213	335	427
0	43 370	39 860	38 570	37 850	37 370
4 580	30 250	28 290	28 340	29 360	29 050
10 600	15 750	15 440	16 730	19 350	20 860
13 700	9 520	9 340	10 140	11 480	12 410
15 200	5 830	5 650	6 230	6 940	7 300

Altitude, m	Combat rated thrust for velocity, m/sec, of -				
	30.5	122	213	335	427
0	25 580	26 640	30 830	39 410	46 480
4 580	17 080	18 680	22 370	31 670	41 770
10 600	8 750	9 920	12 140	18 680	25 040
13 700	4 670	5 560	7 210	11 830	15 390
15 200	2 670	3 250	4 180	6 850	9 160

TABLE III.- CONTROL LAW 2 GAIN SCHEDULE

M	K_q, per sec, for h, m, of -		
	0	6100	12 200
0.4	1.0	1.0	1.4
.7	.7	.7	1.4
1.0	.36	.36	1.4

M	K_θ, for h, m, of -		
	0	6100	12 200
0.4	4.0	4.0	2.85
.7	1.43	1.43	2.85
1.0	2.78	2.78	2.85

M	K_p, per sec, for h, m, of -		
	0	6100	12 200
0.4	-0.715	-0.715	-0.90
.7	-.60	-.60	-.90
1.0	-.40	-.40	-.90

M	K_ϕ, for h, m, of -		
	0	6100	12 200
0.4	2.80	2.80	2.0
.7	2.34	2.34	2.0
1.0	4.5	4.5	2.0

M	K_r, per sec, for h, m, of -		
	0	6100	12 200
0.4	0.285	0.285	0.50
.7	.176	.176	.50
1.0	.12	.12	.50

TABLE IV.- SYSTEM STEP RESPONSE

	Peak value ΔN_z, g units	Steady state q, deg/sec	Steady state p, deg/sec
F-8C plus control system 1 response to 1/2-full-stick deflection step (48-N step)			
M = 0.67; h = 6100 m	1.8	5.0	76
M = 0.90; h = 12 200 m	1.5	5.2	78
F-8C plus control system 2 response to a 22-N stick input step			
M = 0.67; h = 6100 m	3.0	9.2	60
M = 0.90; h = 12 200 m	2.6	9.2	69

TABLE V.- SYSTEM DYNAMIC RESPONSE CHARACTERISTICS

(a) Longitudinal response

M	Altitude, m	Short period: ω, rad/sec	Short period: ζ	Phugoid: λ_1, rad/sec or ω, rad/sec	Phugoid: λ_2, rad/sec or ζ	Actuator: λ, rad/sec
F-8C plus control law 1						
0.67	6 100	4.00	0.80	λ = -0.012	λ = -0.002	λ = -5.40
.90	12 200	5.17	.63	λ = -.063	λ = -.001	λ = -6.57
F-8C plus control law 2 maneuvering (stick out of dead band)						
0.67	6 100	8.88	0.69	ω = 0.055	ζ = 0.100	λ = -1.58
.90	12 200	13.00	.48	λ = -.011	λ = .004	λ = -1.04
Stopping (stick inside dead band)						
0.67	6 100	7.70	0.66	λ = -0.436	λ = -0.0160	λ = -3.25
.90	12 200	11.80	.39	λ = -.575	λ = -.0089	λ = -3.94

TABLE V.- Concluded

(b) Lateral response

M	Altitude, m	Dutch roll		Roll plus control system lag		Spiral	Washout	Trim	Actuator	
		ω, rad/sec	ζ	ω, rad/sec or λ_1, rad/sec	ζ or λ_2, rad/sec	λ, rad/sec	λ, rad/sec	λ, rad/sec	λ_1, rad/sec or ω, rad/sec	λ_2, rad/sec or ζ
F-8C plus control law 1										
0.67	6 100	2.83	0.27	ω = 5.65	ζ = 0.96	-0.0022	-1.16	-0.068	λ = -27.4	λ = 0.229
.90	12 200	3.68	.40	ω = 10.10	ζ = .88	0	-1.12	-.700	ω = 22.8	ζ = .99
F-8C plus control law 2 maneuvering (stick out of dead band)										
0.67	6 100	2.44	0.18	λ = -9.6		-0.0078	-0.53			
.90	12 200	2.54	.36	λ = -16.8		-.0020	-.54			
Stopping (stick in dead band)										
0.67	6 100	2.52	0.19	λ = -7.6		-1.97	-0.51			
.90	12 200	2.56	.36	λ = -14.6		-1.94	-.54			

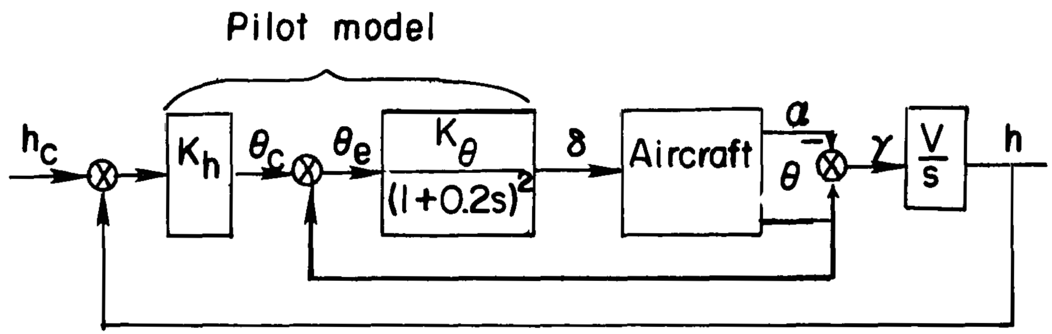


Figure 1.- Pilot aircraft block diagram used for pilot model analysis.

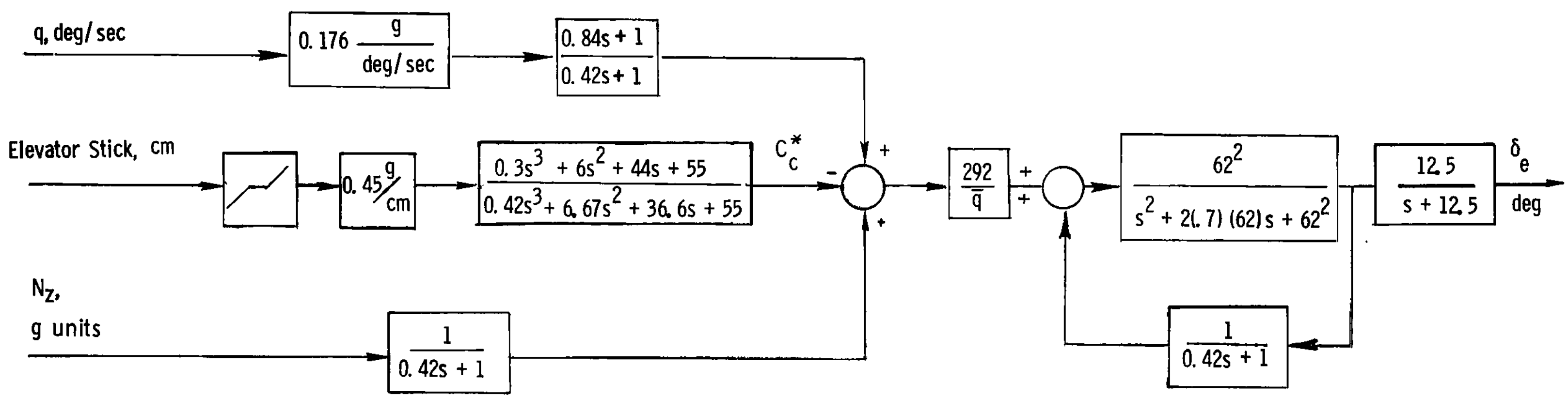


Figure 2.- Longitudinal augmentation system for control law 1.

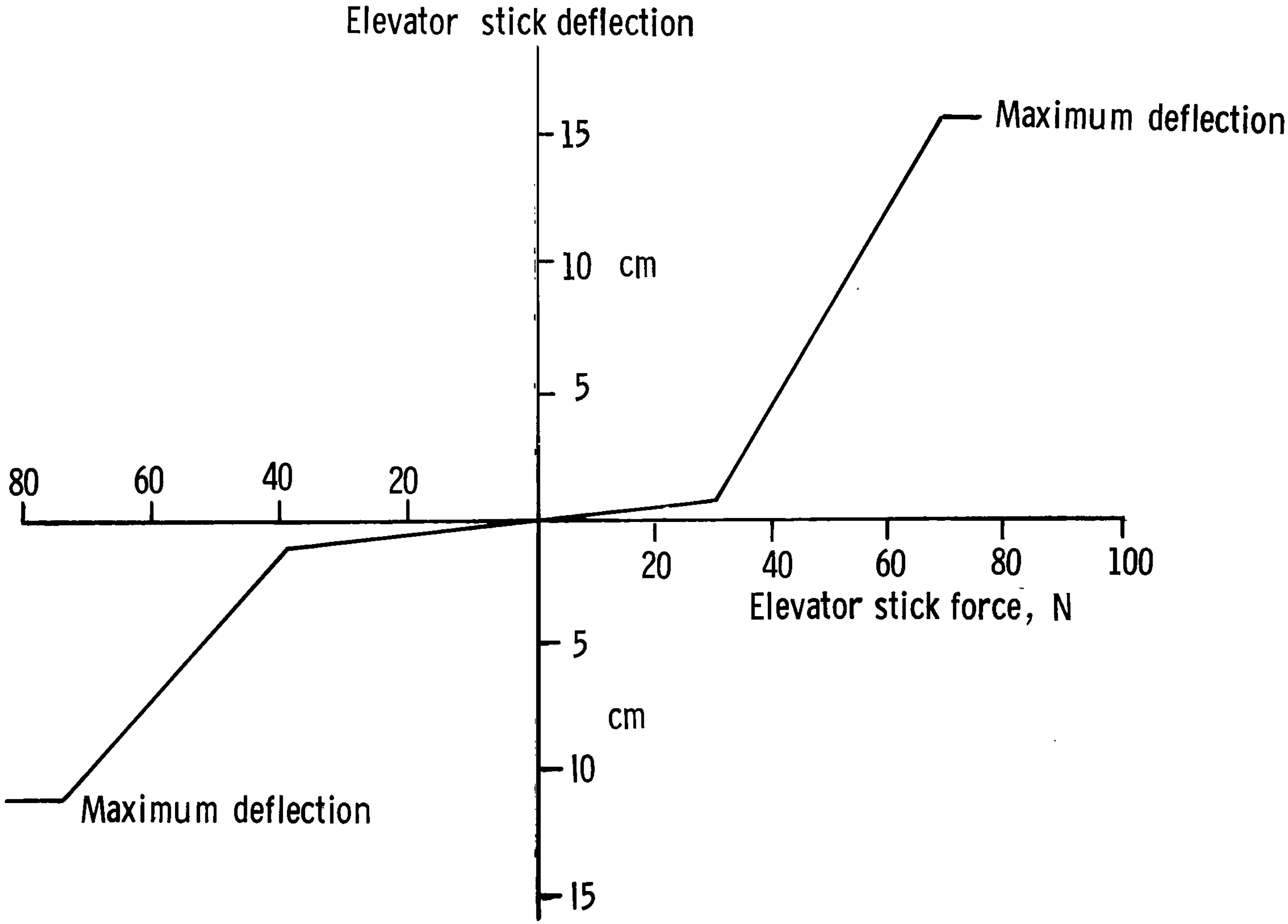


Figure 3.- Elevator stick force characteristics for control law 1.

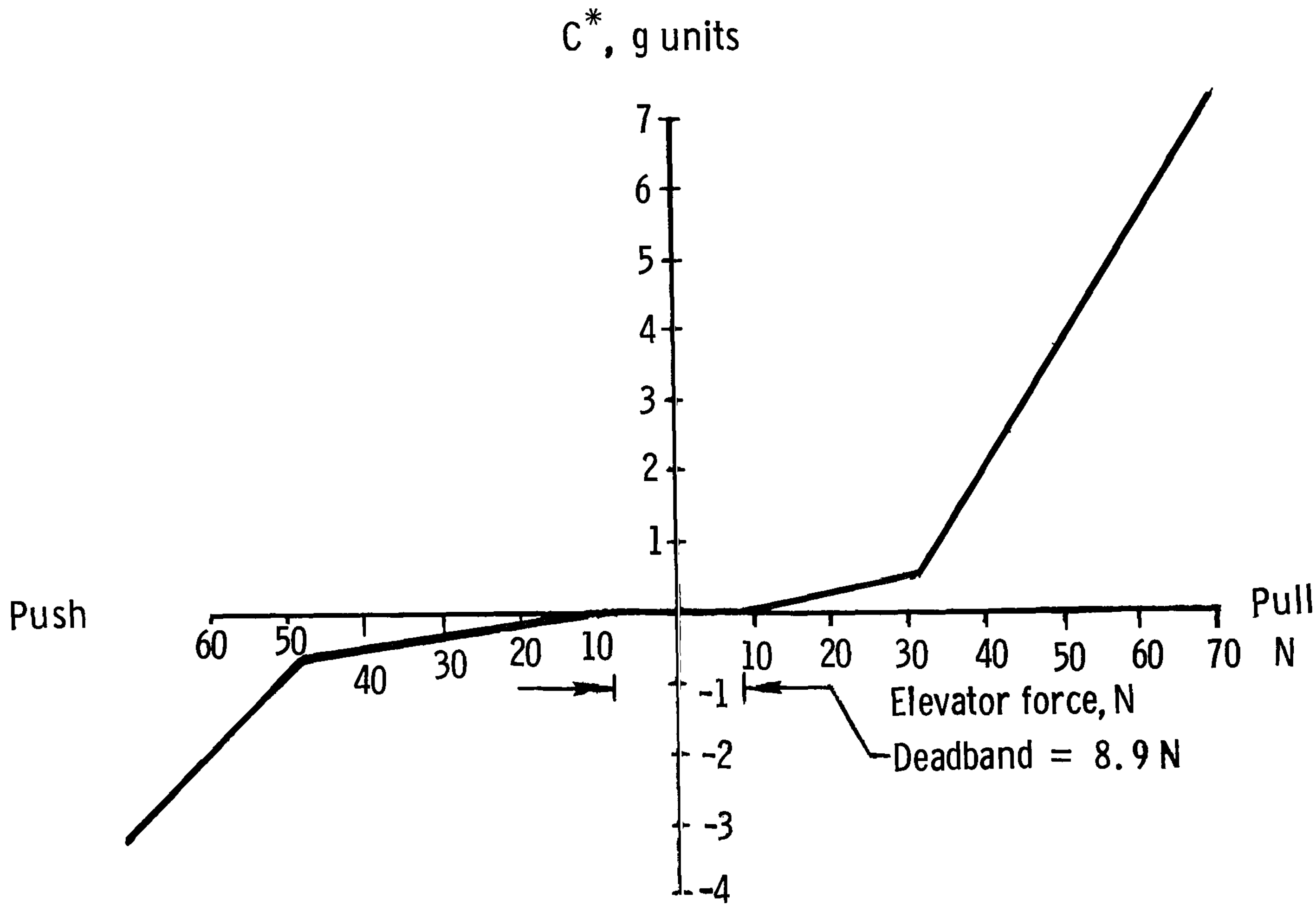
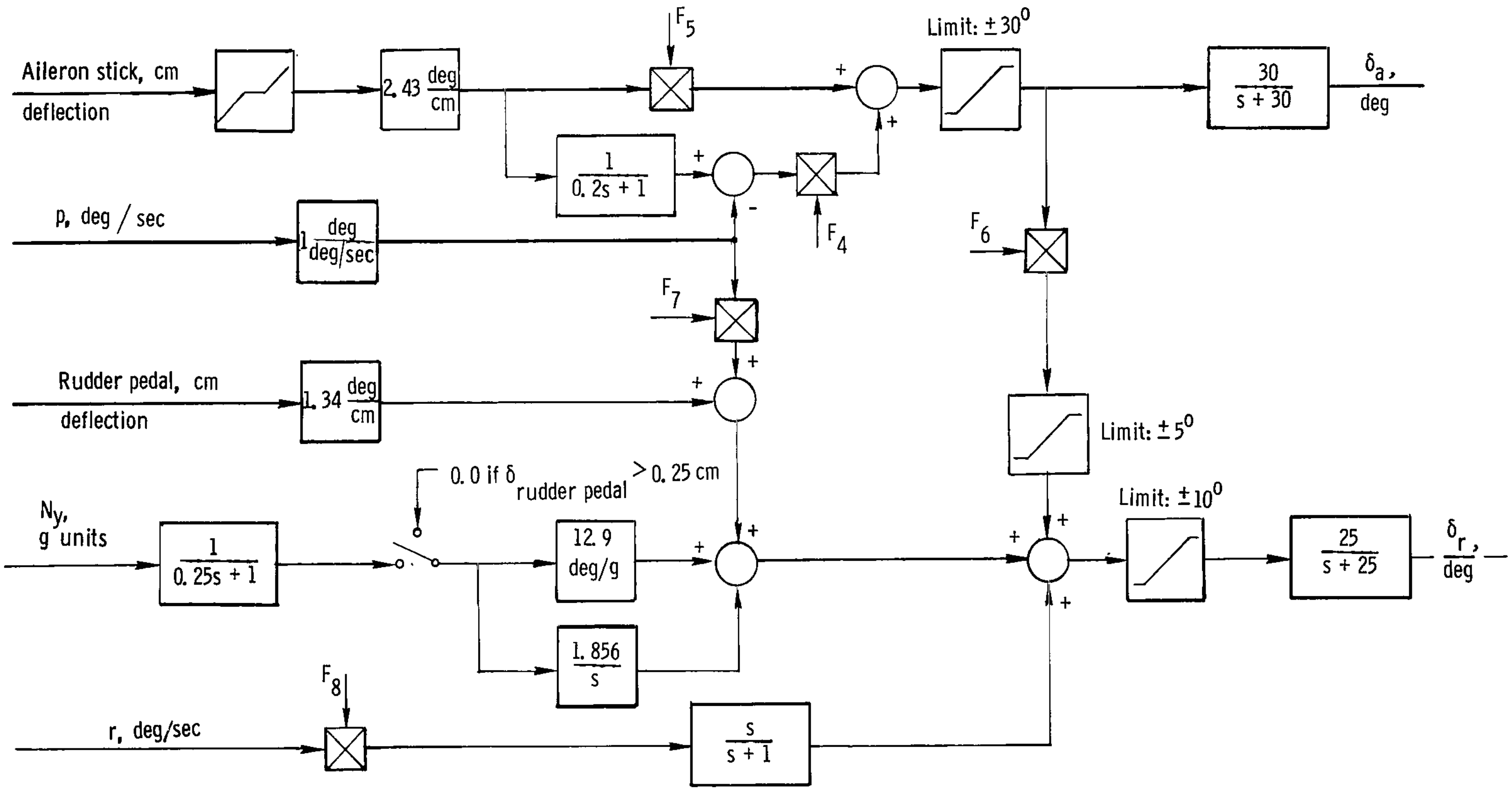
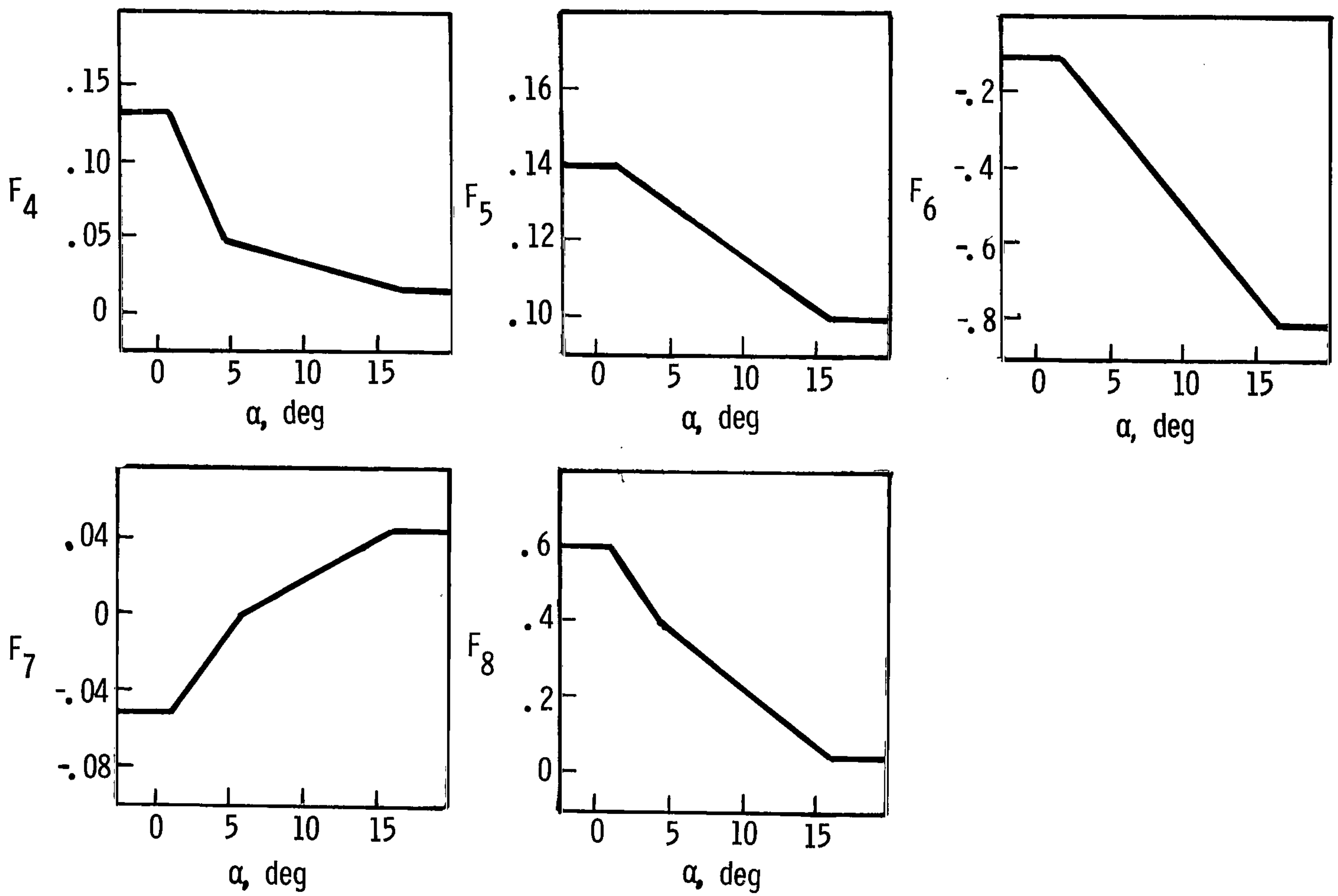


Figure 4.- Elevator stick force and static sensitivity characteristics for control law 1.



(a) Block diagram.

Figure 5.- Lateral augmentation system for control law 1.



(b) Gain schedule.

Figure 5.- Concluded.

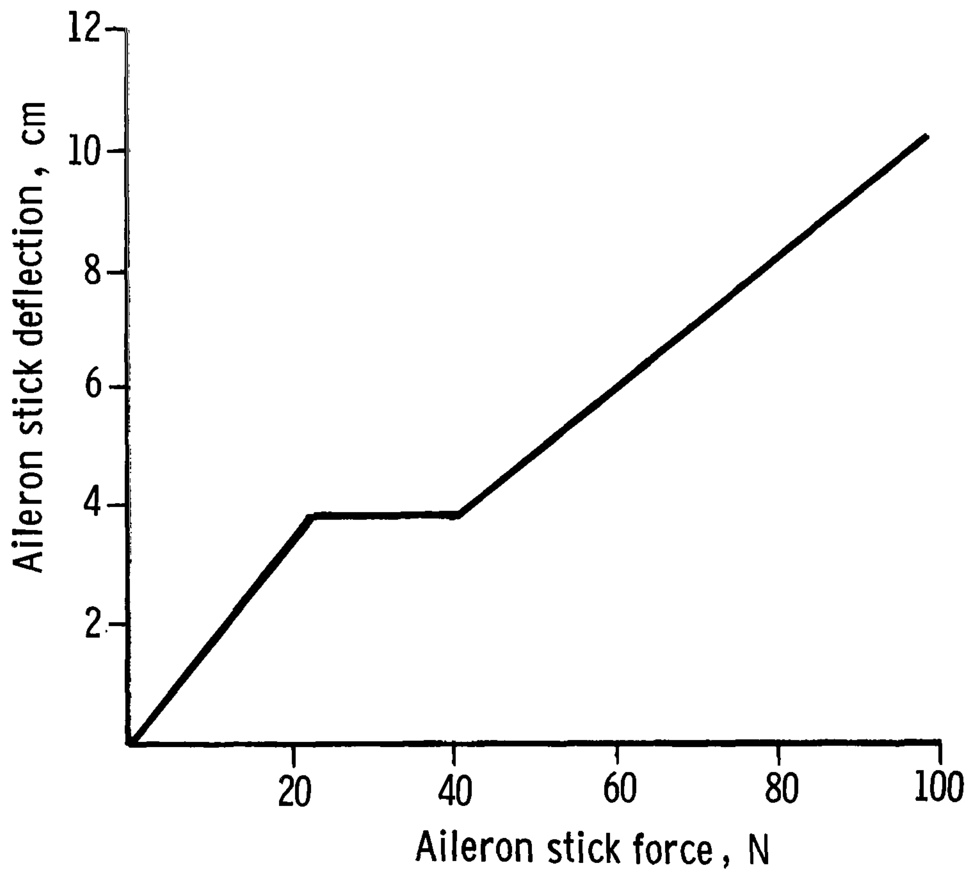


Figure 6.- Aileron stick force characteristics for control law 1.

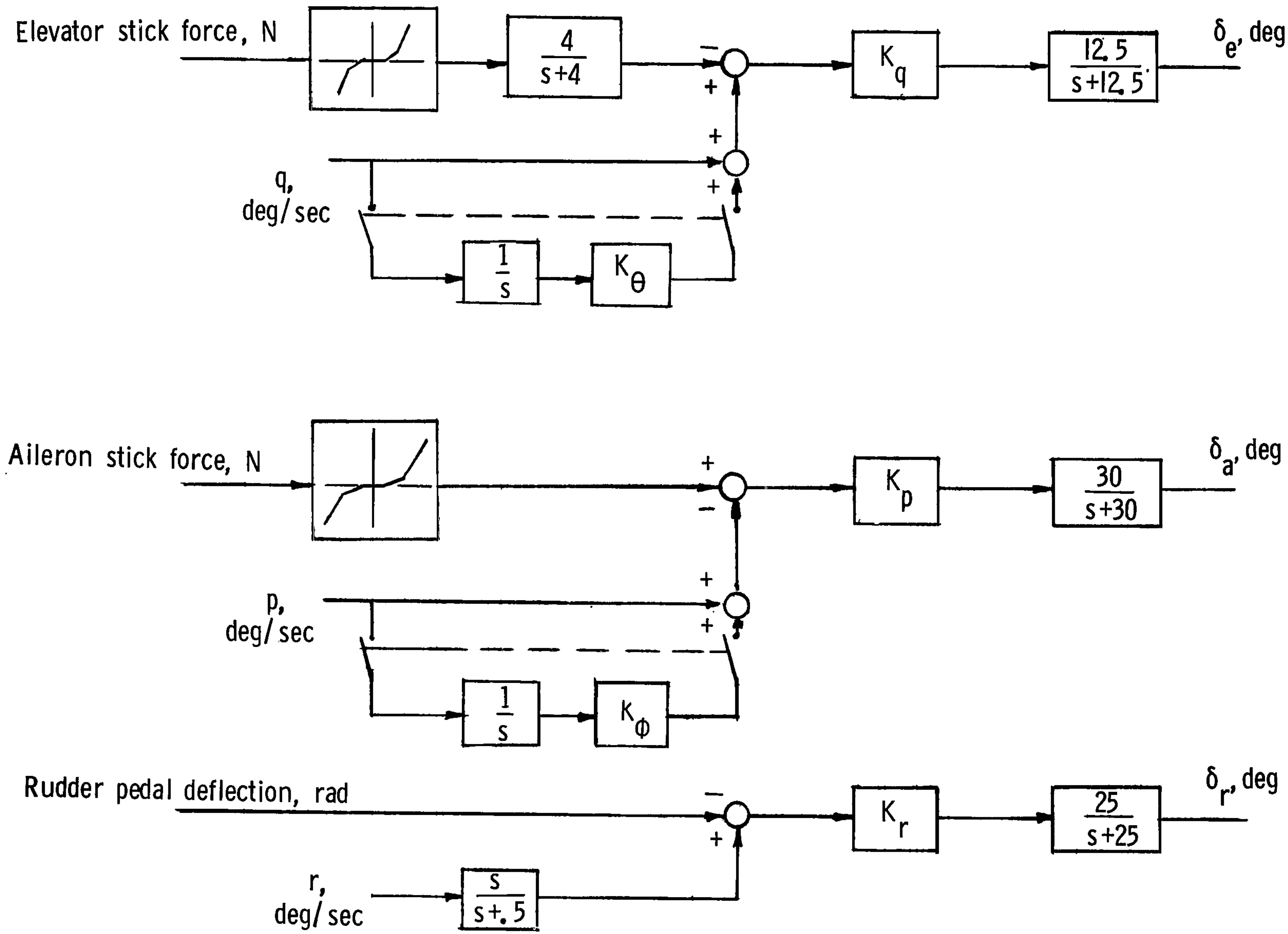
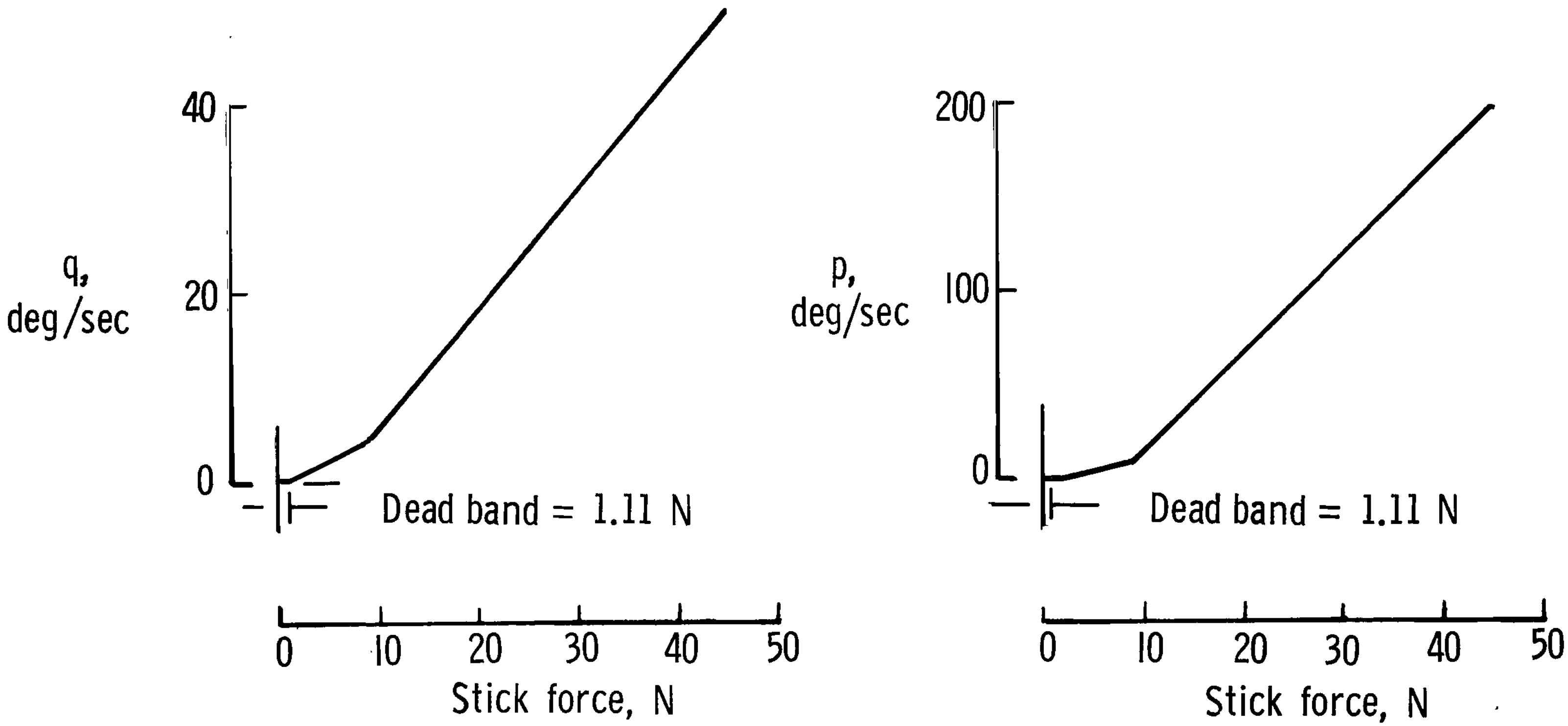


Figure 7.- Block diagram for control law 2.



(a) Longitudinal.

(b) Lateral.

Figure 8.- Control stick characteristics for control law 2.

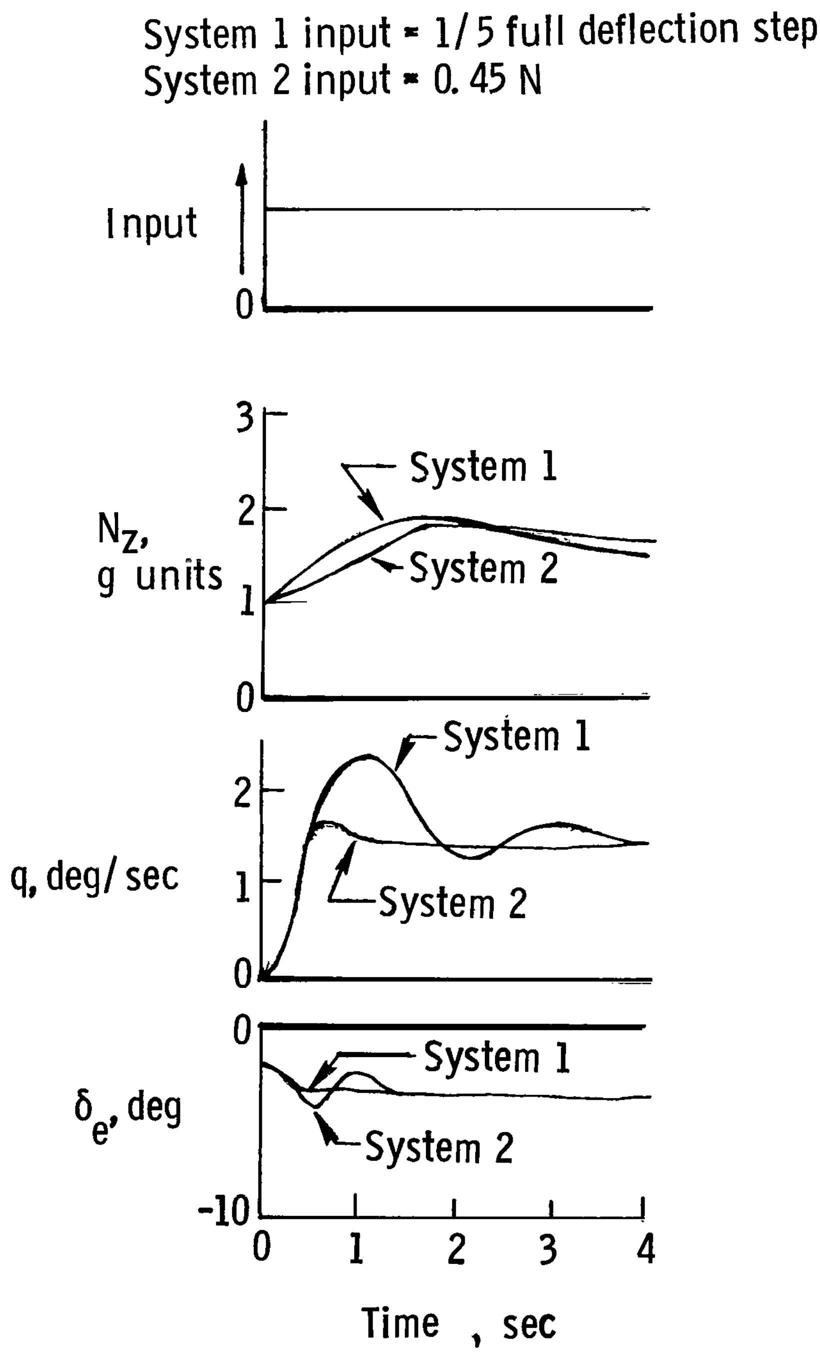


Figure 9.- Control system responses to step control inputs. M = 0.67; h = 6100 m.

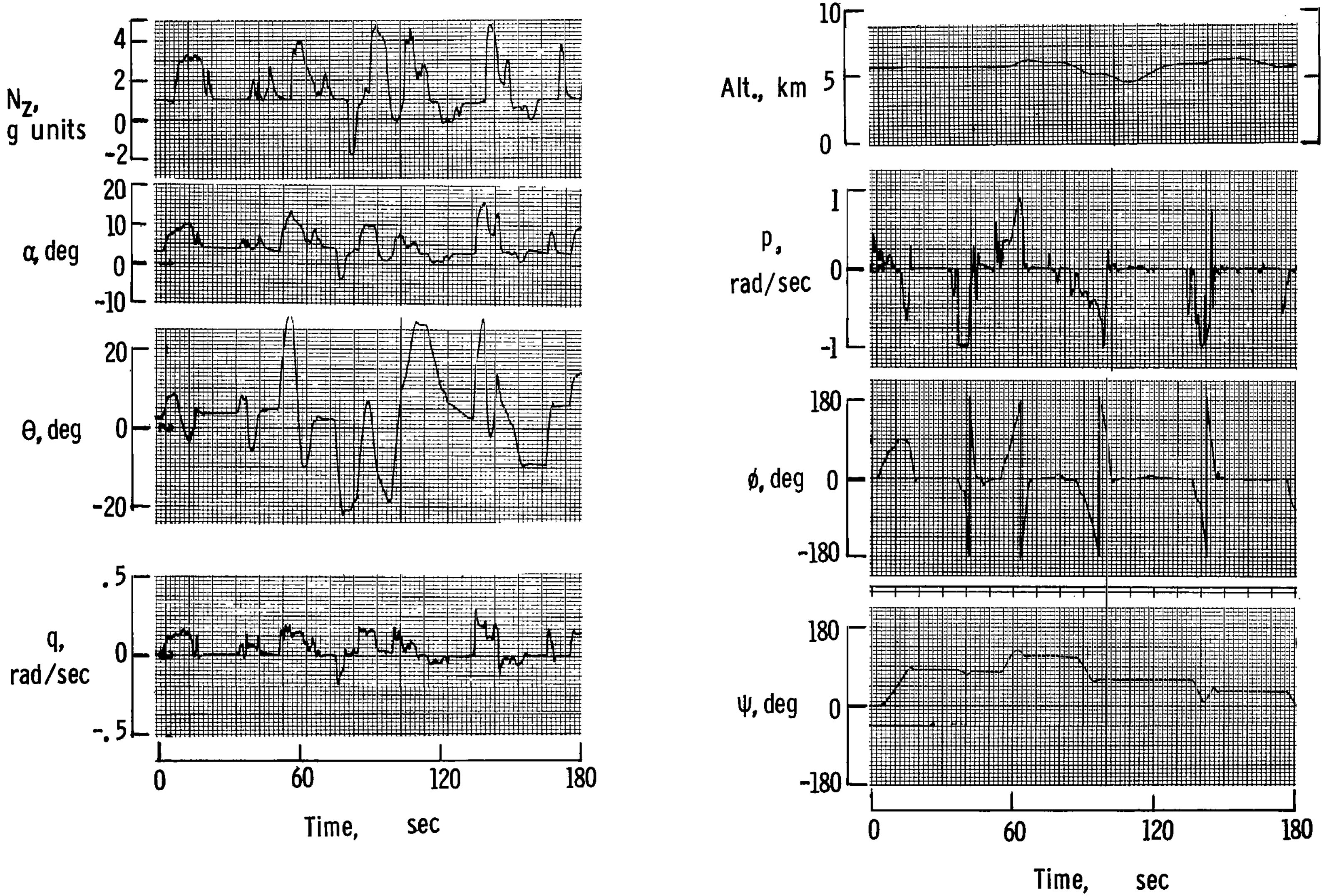


Figure 10.- Target time history.

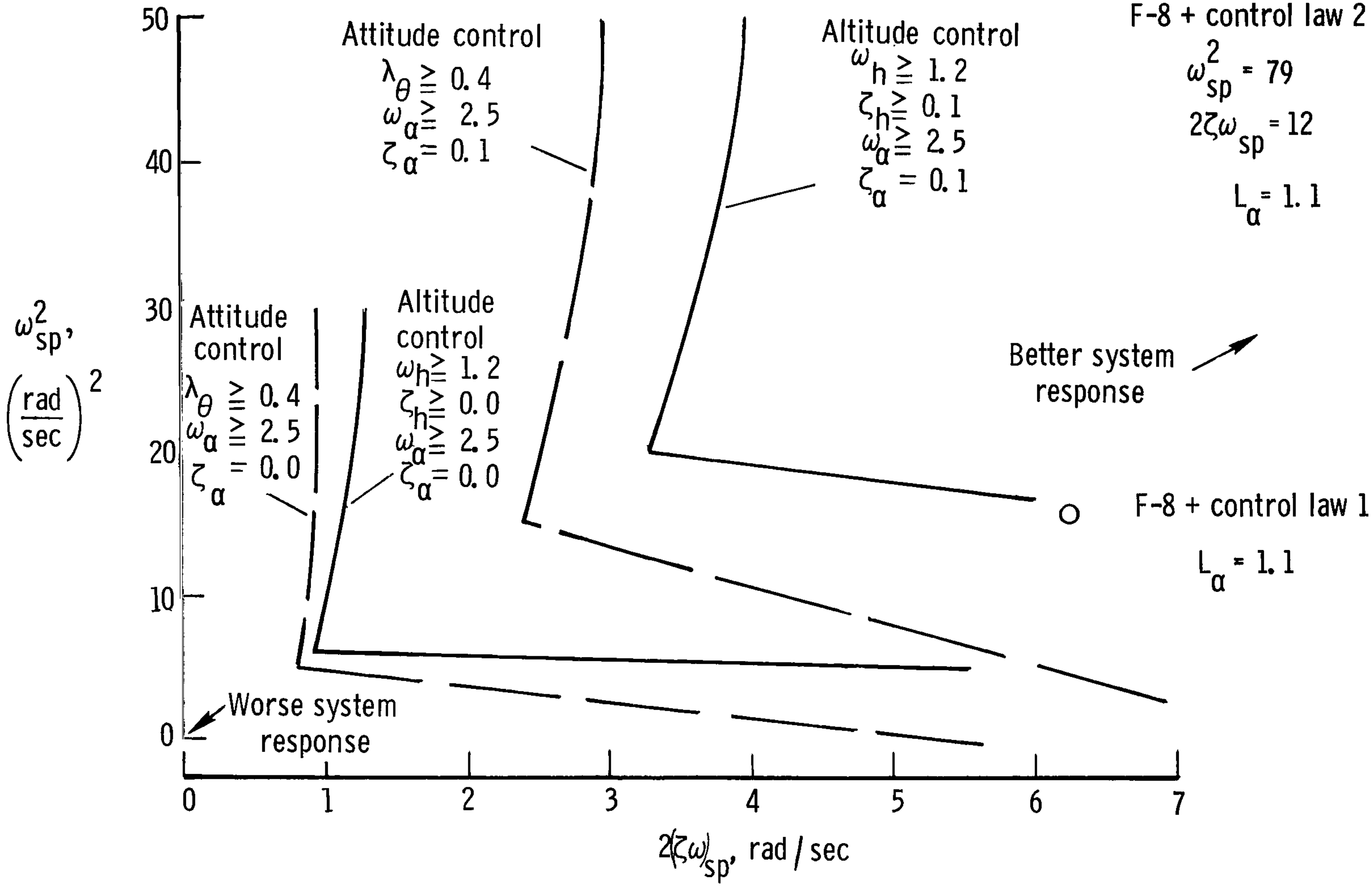


Figure 11.- Boundaries of required aircraft short period characteristics to achieve pilot-aircraft system response characteristics noted in figure; aircraft $L_\alpha = 1.3$. Also noted are aircraft plus control laws 1 and 2 short period characteristics.

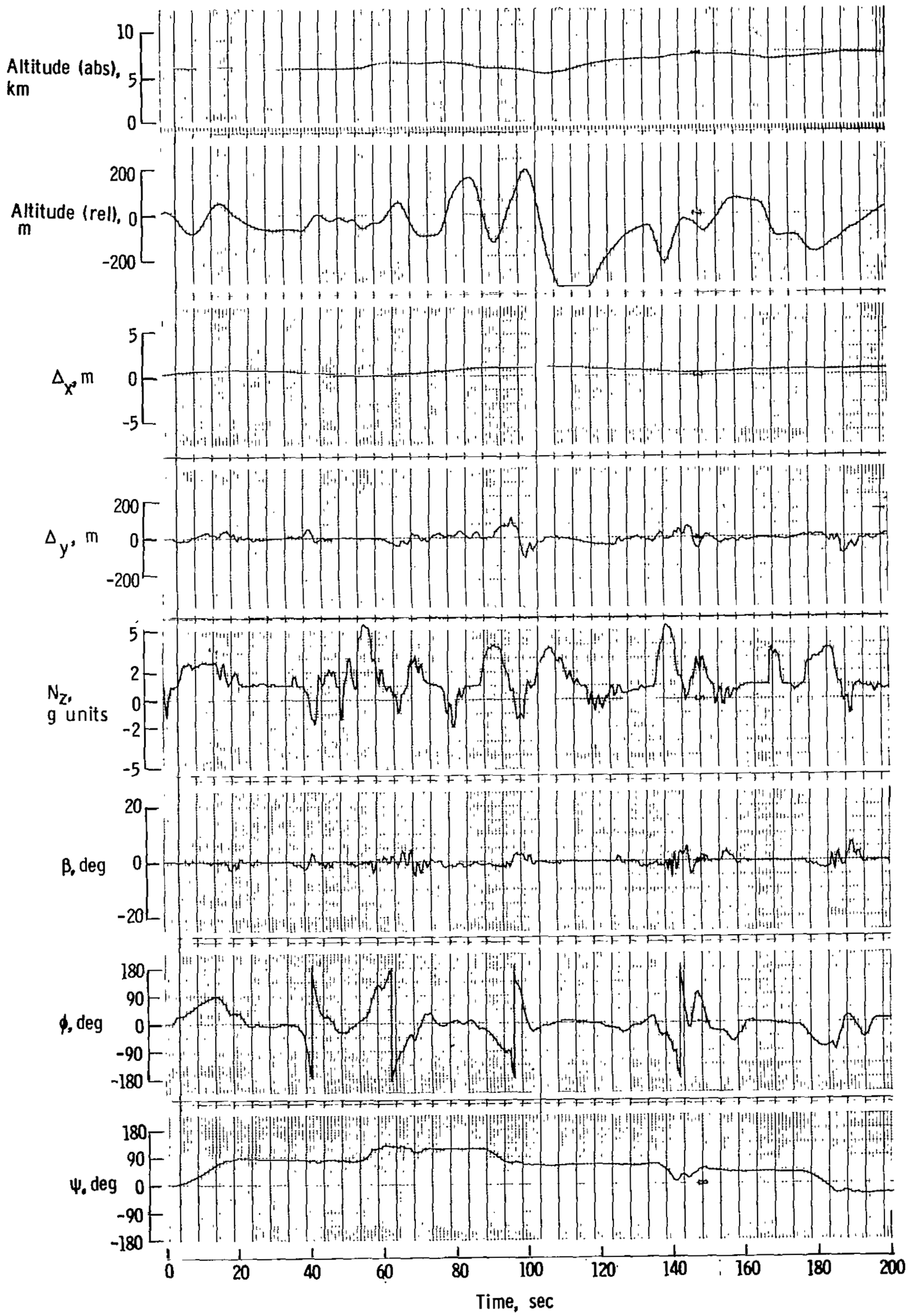


Figure 12.- Target tracking with control law 1.

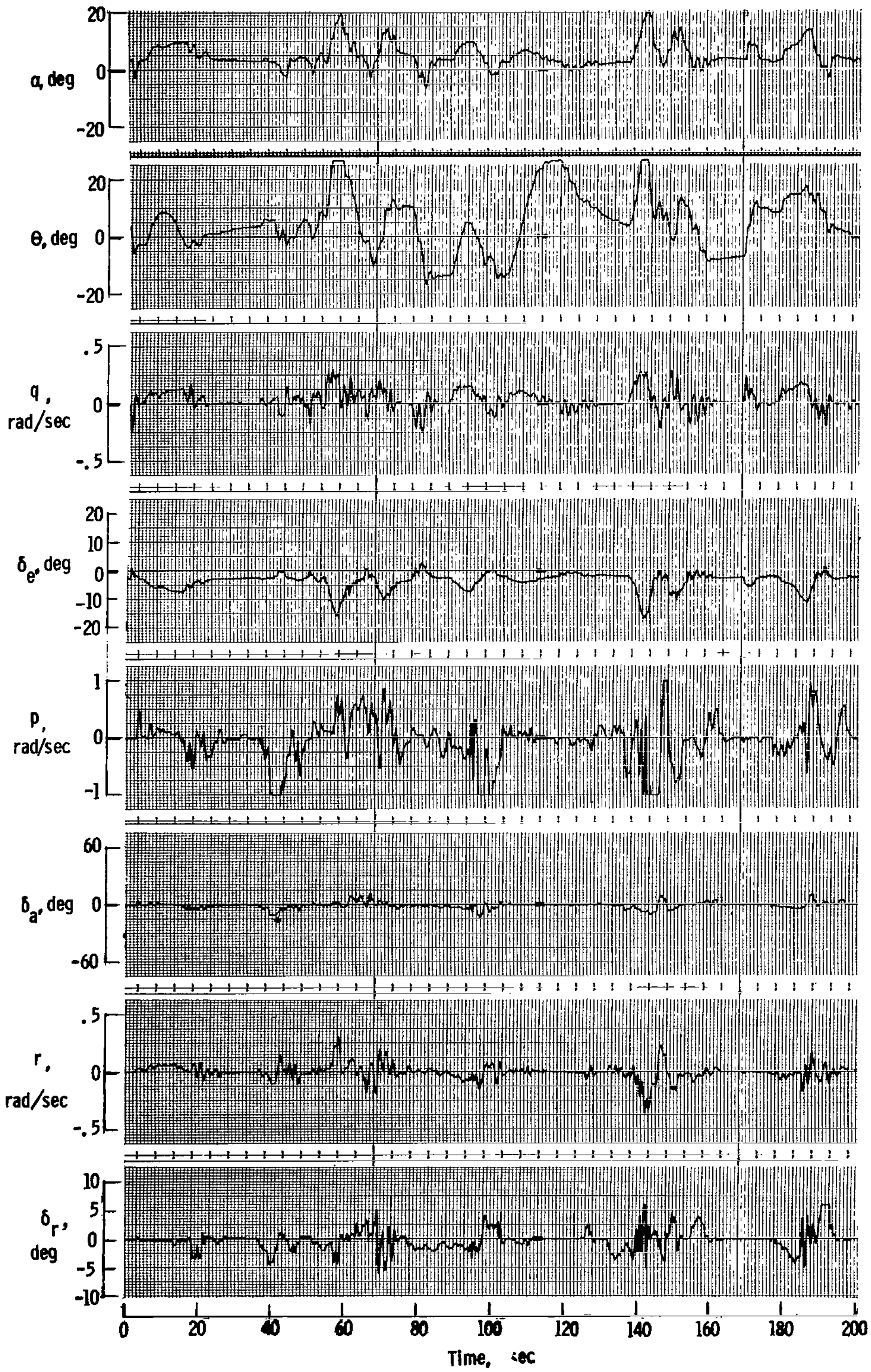


Figure 12.- Concluded.

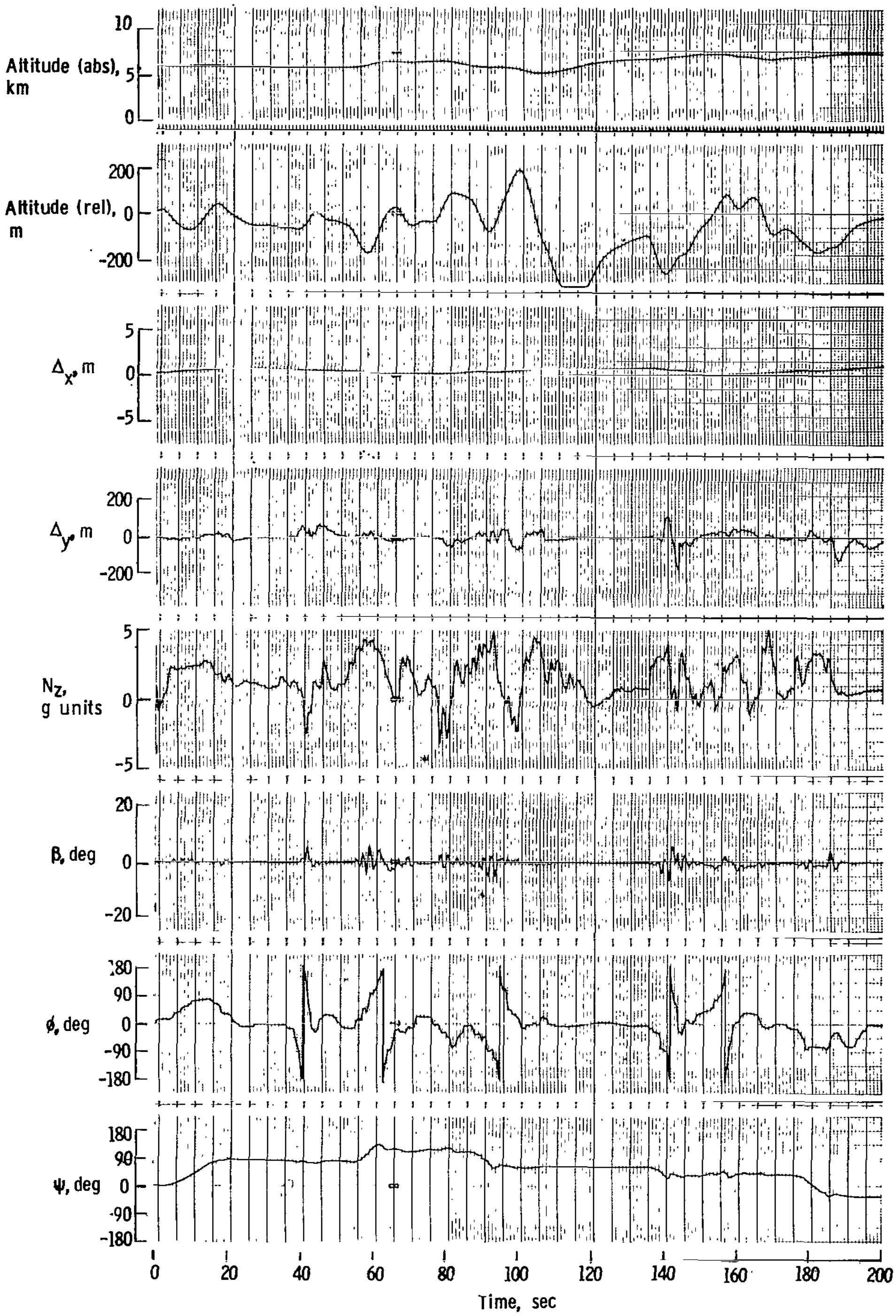


Figure 13.- Target tracking with control law 2.

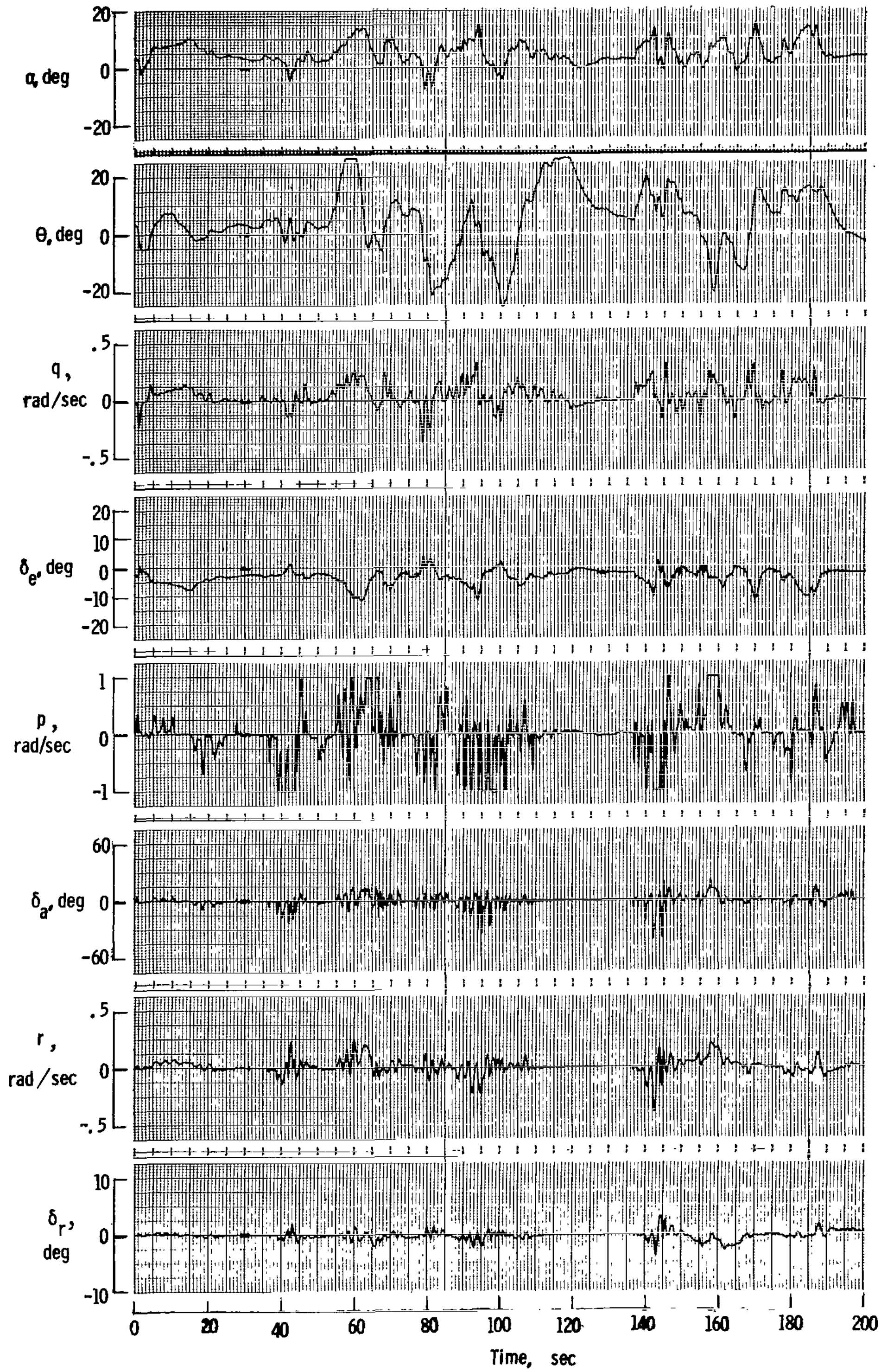
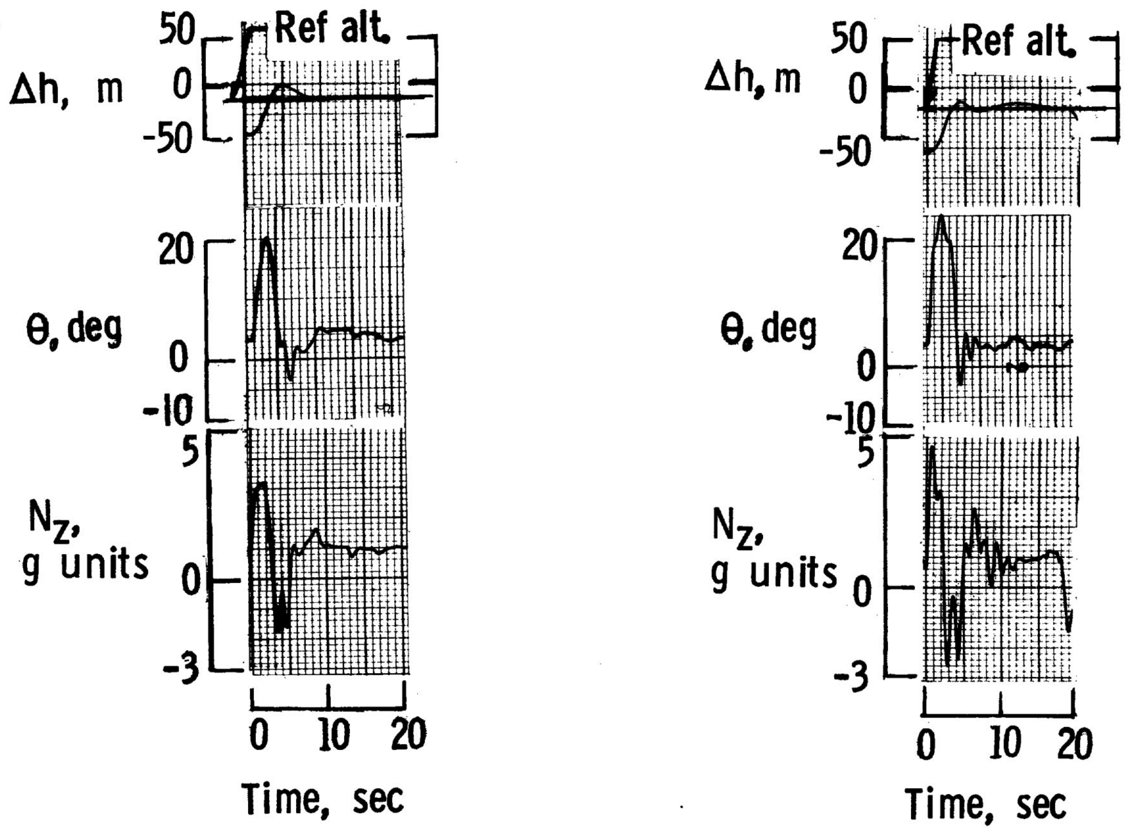


Figure 13.- Concluded.



(a) Control system 1.

(b) Control system 2.

Figure 14.- Step altitude change.

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16. Abstract

A pilot-model analysis has been performed that relates pilot control compensation, pilot-aircraft system response, and aircraft response characteristics for longitudinal control. The results show that a higher aircraft short period frequency is required to achieve superior pilot-aircraft system response in an altitude control task than is required in an attitude control task. These results were confirmed by a simulation study of target tracking. It was concluded that the pilot-model analysis provides a theoretical basis for determining the effect of control task on pilot opinions.

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